EVAPORATIVELY COOLED ROTOR FOR A GAS TURBINE ENGINE

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FIELD OF THE INVENTION

The present invention relates generally to gas turbine engines and, more specifically, to an improved evaporatively cooled rotor for a gas turbine engine.

BACKGROUND OF THE INVENTION

Gas turbine engines, which are used to power aircraft, ships and other vehicles, and to provide shaft power in stationary installations, typically include a compressor section, a combustion chamber and a turbine section. Depending on the particular application, additional flow components including an intake diffuser, power turbine and a nozzle may also be incorporated.

During operation of the engine, air is ingested by the compressor section, which typically includes alternating rows of rotating and stationary blades. After the compressed air exits the compressor, it is typically decelerated for pressure recovery and distributed to one or more combustion chambers. There fuel is injected into the air and the air/fuel mixture is burned at approximately constant pressure.. After combustion, the hot combustion gases are expanded through the turbine section, which typically includes an entry nozzle followed by one or more series of rotating and stationary blade rows for extraction of mechanical work from the expansion of the hot gases. The extracted work is used to rotate a shaft which, in turn, drives the rotating blades of the compressor.

In a basic aircraft propulsion application, the hot gases are then further expanded and discharged through a nozzle at a high velocity, providing substantial thrust to the engine. In addition to turning the compressor blades, the turbine section may include a power section to drive other devices, such as a fan as in a turbofan engine, a propeller as in a turboshaft engine or an electrical generator as in a stationary power system.

The optimal thermodynamic temperature for combustion in a gas turbine engine corresponds approximately to a stoichiometric fuel/air ratio which would result in combustion gas temperatures on the order of 4000° Fahrenheit or higher. Conventional gas turbines, however, typically operate at a much lower turbine inlet temperature, ranging from about 2200° to 3000° Fahrenheit, depending on the specific application of the engine. This limitation in operating temperature is due primarily to the lack of materials for use in the turbine section that are capable of withstanding the high temperatures of stoichiometric or near stoichiometric combustion in addition to the tremendous stresses resulting from the high rotational speeds of the turbine section and the harsh oxidizing environment imposed by the combustion products. As a result, conventional gas turbines often employ expensive high temperature superalloys in the turbine sections and yet still operate at relatively low power per unit of airflow and low fuel efficiencies when compared to the results that could be achieved at higher levels of fuel/air stoichiometry.

Similarly, the development of advanced turbine engines operating at higher pressure ratios results in air temperatures in the high pressure stages of the compressor that may

exceed allowable limits for desirable structural materials. Consequently, such engine designs may require the use of active cooling or expensive superalloys in portions of the compressor section as well.

In an effort to overcome these limitations, substantial efforts have been directed primarily to cooling parts of the turbine section to a temperature substantially below that of the hot combustion gases. Most gas turbine engines, for example, divert a flow of “cool” bleed air from the compressor (bypassing the combustor) to hollow turbine blades. The turbine blades further include a series of pin holes along the blade surface to permit the “cool” bleed air from the blade interior to flow onto the blade surface thereby providing a thin film of cool air over the blade’s external surface. This flow of bleed air then mixes with the expansion flow of hot combustion gases in the turbine section and is discharged through the nozzle of the engine. This type of blade cooling typically lowers the temperature of the turbine blades to approximately 1800° Fahrenheit or less at the conventional operating temperatures noted above.

This approach, however, has many drawbacks. First, the drawing off of bleed air lowers the operating efficiency of the engine. Specifically, since the bleed air used to cool the turbine blades was not burned during the combustion process, it provides much less expansion work in the turbine section than the main combustion gas flow. Nonetheless, the turbine expended work compressing this air. Consequently, an optimum allowable quantity of bleed air is defined by balancing the reduction in the level of expansion work performed in the turbine section with the benefit of higher combustion gas temperature. In addition this approach is typically unable to lower blade temperatures to the point where expensive superalloy materials in the turbine section are no longer necessary.

Other design approaches considered or under development include various forms of forced liquid or gas flow through the turbine blades from an external source. Both closed and open flow paths within the turbine blades have been explored, including water circulation and steam throughflow, similar to cooling with compressor bleed air. These approaches, however, are not attractive for flight vehicles because of the added mass of the coolant. They also introduce added complexity to the engine by requiring coolant delivery from a stationary supply to the rotating components, making them undesirable for stationary or surface vehicle systems as well.

Cooling by fuel circulation through the turbine blades prior to its combustion is also under development. In these designs, heat is absorbed by the fuel as a simple thermal mass, through controlled endothermic decomposition, or by reforming, e.g., of methane with steam. Nonetheless, all of these options require fluid delivery to and removal from the rotating components, as well as specialized fuels, catalysts or thermal conditions.

Another system used to cool turbine blades, which is depicted in FIG. **1** and disclosed in U.S. Pat. No. 5,299,418, the entire disclosure of which is incorporated herein by reference, includes the use of a cooling fluid F disposed within a sealed, internal cavity **34** in each turbine rotor **18**. The rotor **18** includes a hub or disc section **28** proximate to the axis of rotation A—A of the engine and a blade section **30** extending outwardly from and supported by the hub **28**. The hot combustion gases typically flow past the blade section **30**. The rotor **18** further includes a wall **32** which defines the cavity **34**. The cavity **34**, moreover, is divided into a condensing section **36** located at the hub **28** and a vaporization section **38** located out at the blade **30**.